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NASA TN D-2609

THERMAL DESIGN AND THERMAL DATA ANALYSIS OF SA-5 PAYLOAD

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1965

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THERMAL DESIGN AND THERMAL DATA ANALYSIS OF SA-5 PAYLOAD

SUMMARY

The SA-5 payload was injected into orbit on January 29, 1964. The payload consisted of the S-IV stage, the IU (instrument unit), the adaptor, and a Jupiter nose cone. Inside the adaptor was a beacon transmitter whose power supply was batteries located in the nose cone. Temperature specifications on the batteries and the transmitter required thermal control in orbit. The specifications were as follows:

Beacon transmitter range	-10 to 75 $^\circ$C
Nose cone batteries range	-20 to 75 $^\circ$C

The in-flight temperature measurements (of these two components) show that the temperatures remained within the following ranges:

Beacon transmitter range	-4 to 13 $^\circ$C
Nose cone battery range	4 to 20 $^\circ$C

Presented in this report are the thermal requirement of the SA-5 payload, the thermal design, the flight data, and post-launch analysis.

I. INTRODUCTION

Saturn launch vehicle SA-5 (Fig. 1) was launched at 11:25 a.m., EST, on January 29, 1964, from Saturn Launch Complex 37B, Atlantic Missile Range, Cape Kennedy, Florida. This vehicle was the fifth to be flight tested in the Saturn I R&D program, and is the first of the Block II series.

The SA-5 orbital payload (Fig. 2) consisted of a modified Jupiter nose cone and aft unit with an adaptor section which was mated to the

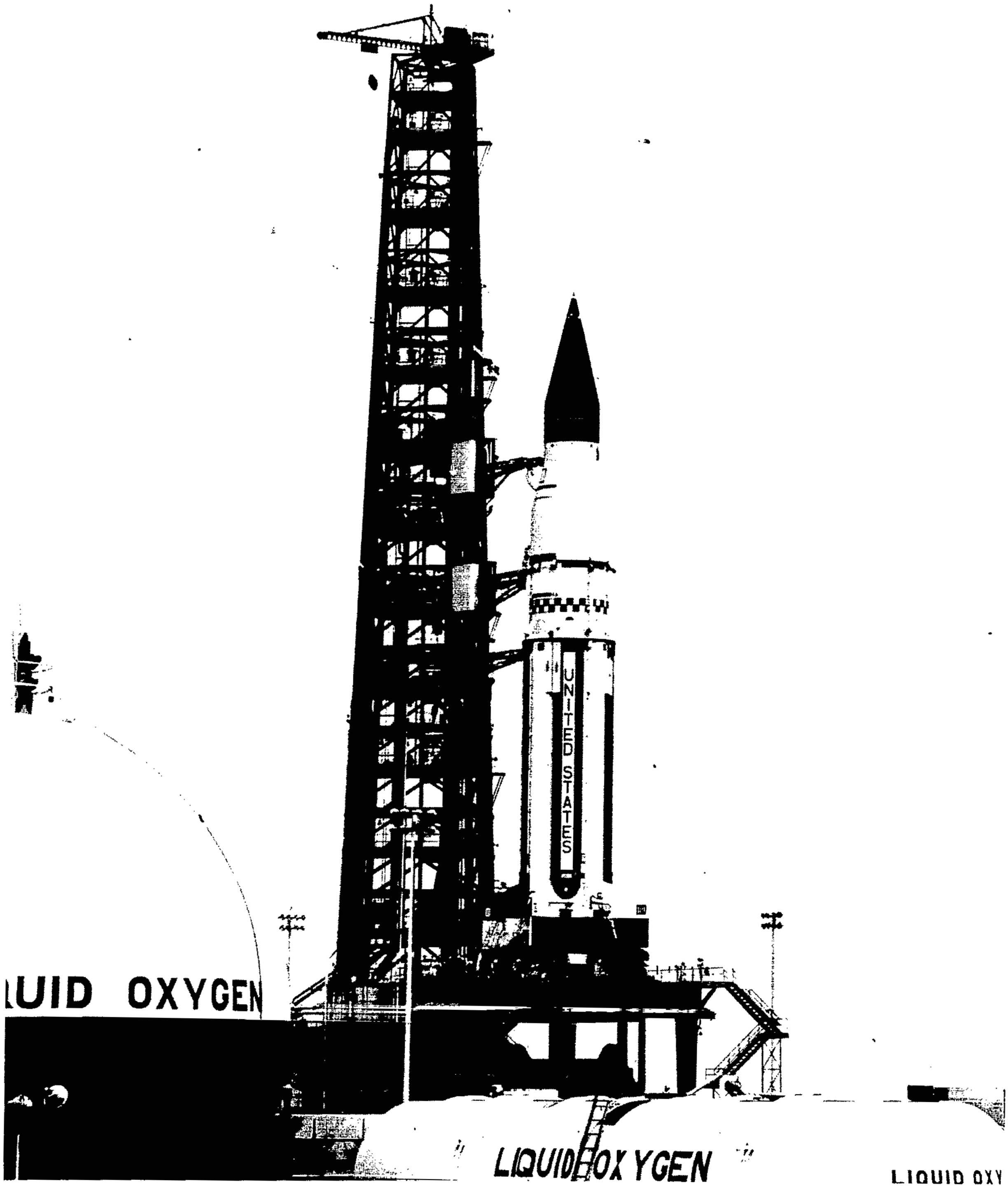


FIGURE 1. SA-5 VEHICLE ON THE PAD AT LAUNCH COMPLEX 37B

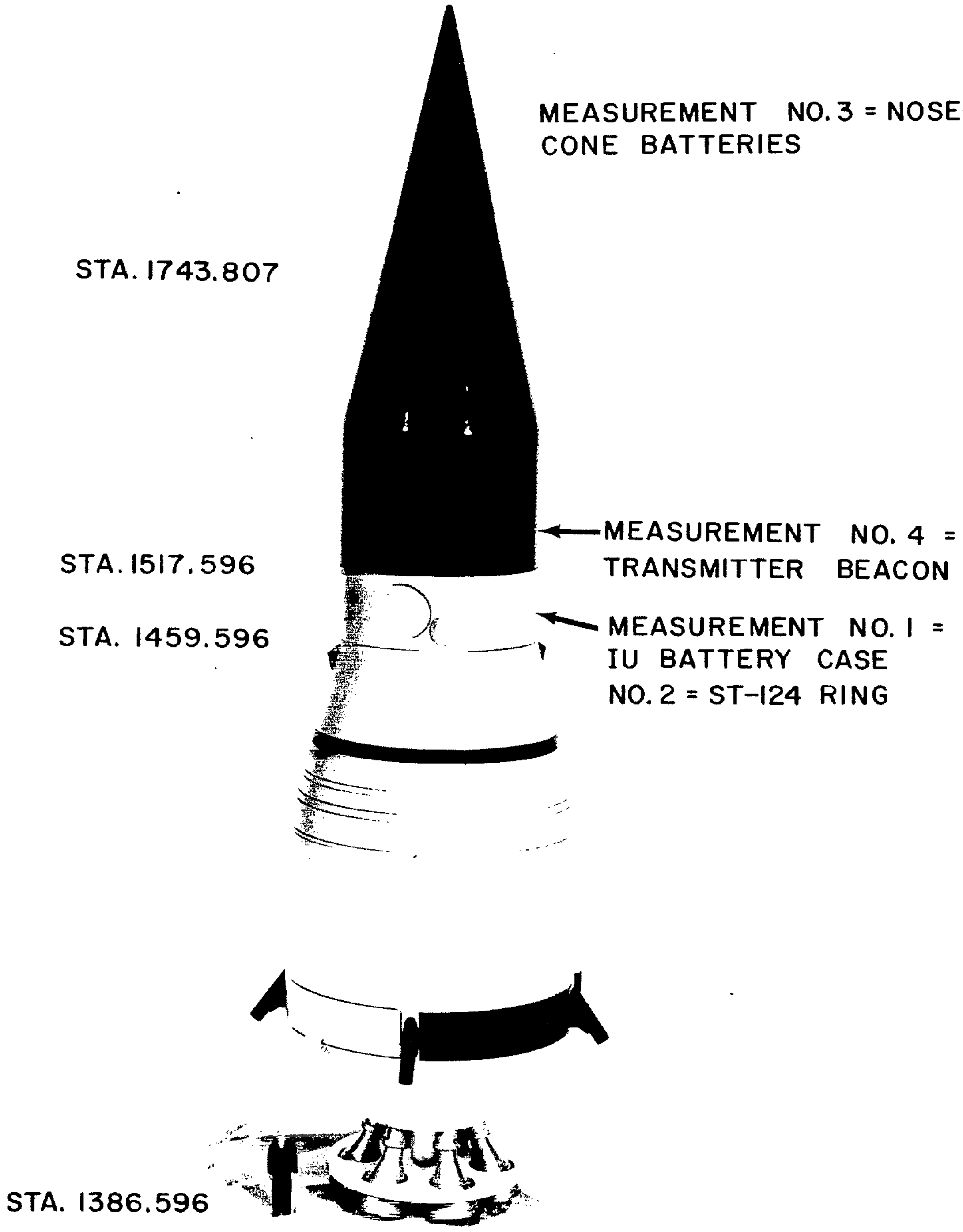


FIGURE 2. SA-5 PAYLOAD MODEL

instrument unit and S-IV stage. This assembly (excluding the S-IV) was approximately 37 feet long and 13 feet in diameter (of the IU base). Sand ballast simulated the mass characteristics of the Apollo payload in the SA-5 payload. The orbital insertion elements [1] are as follows:

Time of orbital insertion after launch (sec)	639.56
Apogee altitude (km)	771.40
Perigee altitude (km)	262.30
Period (min)	94.86
Inclination (deg)	31.46
Pitch angle (deg)	89.74
Altitude (km)	262.4
Eccentricity	.0356
Radius of perigee (km)	6645
Nominal lifetime (days)	451

Inside the adaptor was a beacon transmitter whose power supply consisted of batteries located in the nose cone. The beacon transmitter was to be active at least five weeks. Each of these components has rigid temperature specifications for proper operation to be assured. Thermodynamic studies were performed (utilizing a thermodynamic model) to arrive at the proper thermal design. Reliability and simplicity were considered.

Also, there were four thermocouples located in the payload, one on the transmitter and one on the batteries. The other two were located in the IU 180° circumferentially apart. The Goddard minitrack network transmitted and received the real time data. The transmission of data ceased on April 3, 1964, sixty-five days after launch. Analysis of this data was performed to verify the thermal design and to obtain attitude data.

II. THERMAL REQUIREMENTS

Specifications for the temperature-sensitive electronic components were received as follows:

Component	Mass	Approximate Dimensions	Heat Dissipation Rates	Specified Temperature Operating Range
Beacon transmitter	.68 kg (1.5 lbm)	$13 \times 25 \times 5\ cm^3$ ($5 \times 10 \times 2\ in.^3$)	1.25 to 2 w	-10 to 75° C or 263 to 348° K
Transmitter electronics	1.2 kg (2.6 lbm)	$11 \times 8 \times 4\ cm^3$ ($4.5 \times 3 \times 1.5\ in.^3$)	.3 to .75 w	-15 to 80° C or 258 to 353° K
Nose cone batteries	45 kg (100 lbm)	—	.2 w	-20 to 75° C or 253 to 348° K

III. PREFLIGHT THERMAL STUDIES, THERMAL DESIGN, AND PREDICTED TEMPERATURES

A. TECHNIQUE

Utilization was made of "The General Space Thermal Program" which simultaneously solves a set of n calorimetric equations of the general form given here by numerical iteration.

$$
\begin{aligned}
\dot{T}_i H_i = {} & A_{1i}\, \alpha_i\, S + A_{2i}\, \alpha_i\, SB \\
& + A_{3i}\, \epsilon_i\, SE - A_{4i}\, \epsilon_i\, \sigma T_i^4 \\
& + \sum_{j=1}^{n} \left[C_{ij} (T_j - T_i) + R_{ij} (T_j^4 - T_i^4) \right] \\
& + \dot{Q}_i
\end{aligned}
$$

where

T_i = temperature of node i

$\dot{T}_i = \frac{dT_i}{dt}$

H_i = heat cap of node i

C_{ij} = conductance constant between nodes i and j

R_{ij} = radiance constant between nodes i and j

$\dot{Q}_i$ = internal heat generation of node i

α_i = solar absorptance of node i

ϵ_i = IR emittance of node i

S = solar flux at one A.U.

B = ratio of maximum albedo flux to S

E = ratio of maximum Earth IR flux to S

σ = Stefan-Boltzmann constant

A_{1i} = area function for incident solar energy to node i

A_{2i} = area function for incident albedo energy to node i

A_{3i} = area function for incident Earth IR to node i

A_{4i} = external radiating area of node i

B. SA-5 PAYLOAD THERMODYNAMIC MODEL

The mathematical model (thermal schematic) of the SA-5 payload utilized in this study is shown in Figure 3. The "floating" parameters were varied because (1) their values were controllable depending on the thermal design, (2) their values varied as a function of the orbital perturbations, and/or (3) their values were only known to be within a given range, which was large, necessitating bracketing.

The following predicted orbital parameters were used for the thermal calculations:

(1) Payload in flat tumble with spin rate greater than 0.5 r.p.m.

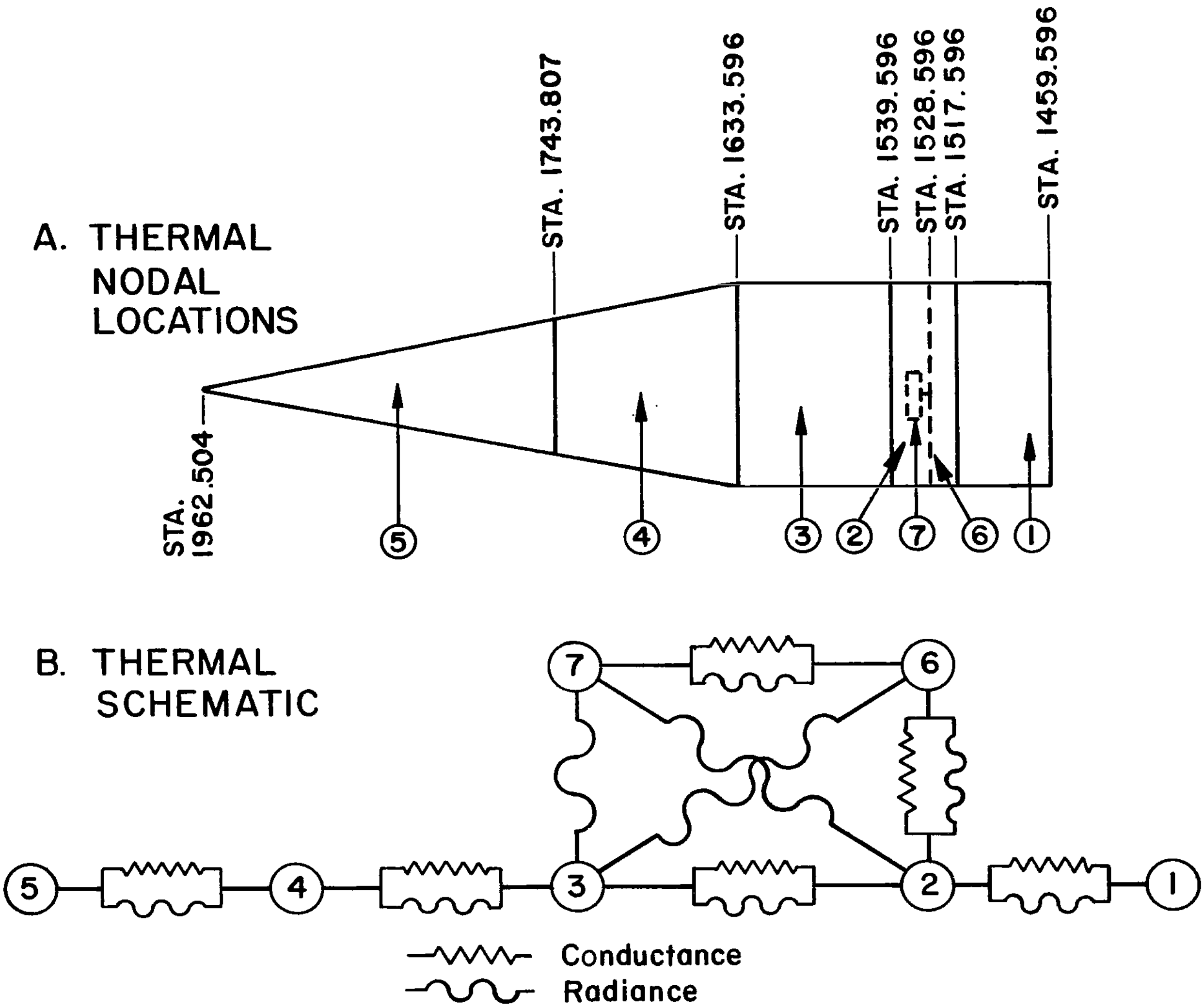
(2) Eccentricity of orbit = 0

(3) Inclination of orbit = 30 degrees

(4) Altitude of orbit = 200 km

(5) Percent time in sunlight (less than 80% and greater than 60%)

(6) All solar angles possible



C. LEGEND

1 IU
2 Part 1 of the adaptor
3 Part 2 of the adaptor
4 Aft
5 Nosecone and batteries
6 Plate
7 Transmitter beacon and electronics

FIGURE 3. SA-5 THERMAL MODEL

C. THERMAL DESIGN

The component temperatures in orbit are a function of internal heat generation, thermal linkage to the payload structure, and structure temperature. The structural temperatures are a function of the conductive and radiant heat transfer to the remainder of the vehicle and the radiant heat transfer to space. Usually the most readily controllable parameter affecting the thermal design is the optical properties of the vehicle surfaces, especially when the component heat generation is low. Studies were performed to determine if thermal control could be obtained by chosing the solar absorptance, α, and infrared emittance, ϵ, of the vehicle surface. The "General Space Thermal Program" was adapted to the thermodynamic model. A series of computer runs were made in which the various "floating" parameters were chosen such that "hot" and "cold" thermodynamic cases were obtained. It was found that thermal control could be achieved if the exterior surface of the SA-5 payload could be coated such that the solar absorptance/infrared emittance ratio, α/ϵ, was 1.3. A coating (readily applicable to large areas) was not available with this ratio. A space-stable black paint (Sherwin-Williams Kem Lustral flat black enamel F-65B2 applied 4 ± 1 mil) was chosen whose α/ϵ is 1.0. This means the temperature will remain cool, but within the temperature specifications of the components.

The exterior coating of the IU was not known to be space stable. A solar absorptance of 0.3 and an IR emittance of 0.9 were used in the preflight calculations. These values were chosen on the basis that they were realistic and conservative.

D. PREDICTED TEMPERATURES OF SA-5 PAYLOAD

The resulting temperature ranges that were obtained are as follows:

Nose cone batteries (T_5) - 277 to 310° K

Beacon transmitter (T_7) - 279 to 303° K

IU (T_1) - 240 to 260° K

IV. FLIGHT TEMPERATURE DATA AND ANALYSIS

The SA-5 payload (Fig. 2 mated to the S-I stage in Fig. 1) was placed in orbit with virtually no attitude deviations and small angular rates (less than 0.1 deg/sec). The ambient roll rate was approximately 14 deg/sec. At the end of lox venting (which ceased after the first orbital pass) and LH_2 venting (which was completed at least within 24 hours), the vehicle was spinning at a rate of approximately 17 deg/sec and had a gyroscopic processional motion with a period of 120 seconds (3 deg/sec equivalent tumble rate), and a half cone angle of 75 degrees.

The SA-5 payload telemetered four temperature measurements (Fig. 2):

Measurement #1 - IU battery case

Measurement #2 - ST-124 mounting ring in the IU

Measurement #3 - Nose cone batteries

Measurement #4 - Transmitter beacon

Figure 4 is a prelaunch example of the SA-5 minitrack telemetry (calibrated at MSFC) showing positions of the four temperature measurements during one cycle. Using a Gerber Variable Scale (a linear interpolator), the telemetry was measured in c.p.s. and converted to °K utilizing the conversion chart in Figure 5. The function $\Delta T/\Delta f$ versus frequency was obtained by numerical differentiation of the $T = f$ (frequency) curve. This curve ($\Delta T/\Delta f$ versus f shown in Fig. 5) was useful in approximating the temperature error. It was estimated that the accuracy for the linear interpolation of the telemetry to frequency is ± 1 c.p.s. With this estimated ΔF, ΔT can be obtained at any given frequency, e.g., at $f = 764$ c.p.s., $T = 292$°K, which gives a $\Delta T \cong \pm 1.1$° K.

Orbital tracking of SA-5 was conducted by the NASA Space Tracking and Data Acquisition Network (STADAN), which is composed primarily of the global network of minitrack stations. Green Mountain Propagation Studies Test Facility (MSFC) also produced data throughout the telemetry lifetime of SA-5. The data were telemetered for 65 days after launch (predicted lifetime was 45 days) until the beacon transmitter ceased functioning on April 3, 1964.

Figure 6 presents the SA-5 temperature data received from Green Mountain Propagation Studies Test Facility. Several orbital passes were tracked per week except for the interval of March 1 through March 9, 1964. The telemetry was recorded on six-channel sanborn strip charts and reduced, as previously described, to temperature (°K)

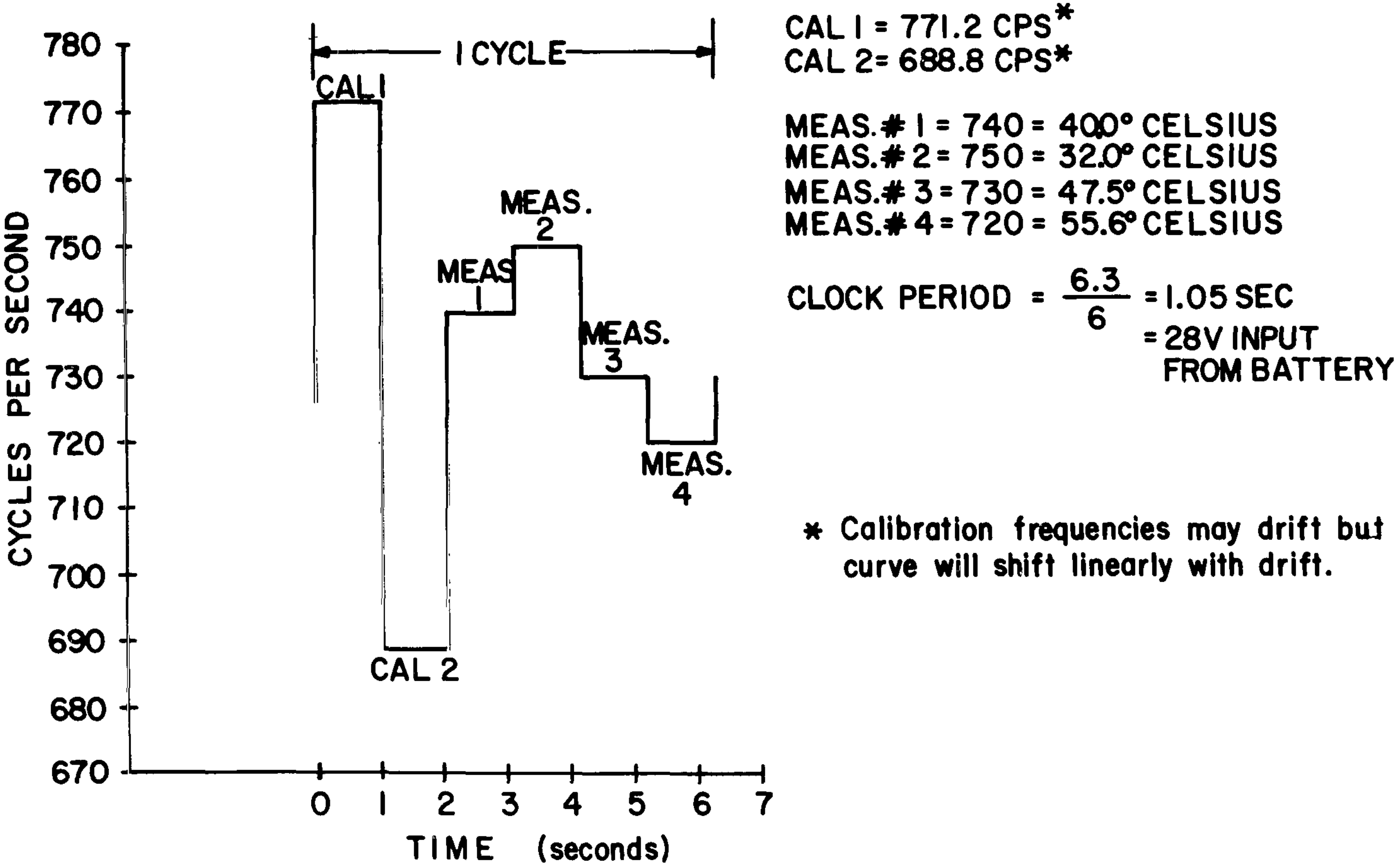


FIGURE 4. EXAMPLE OF SA-5 MINITRACK TELEMETRY

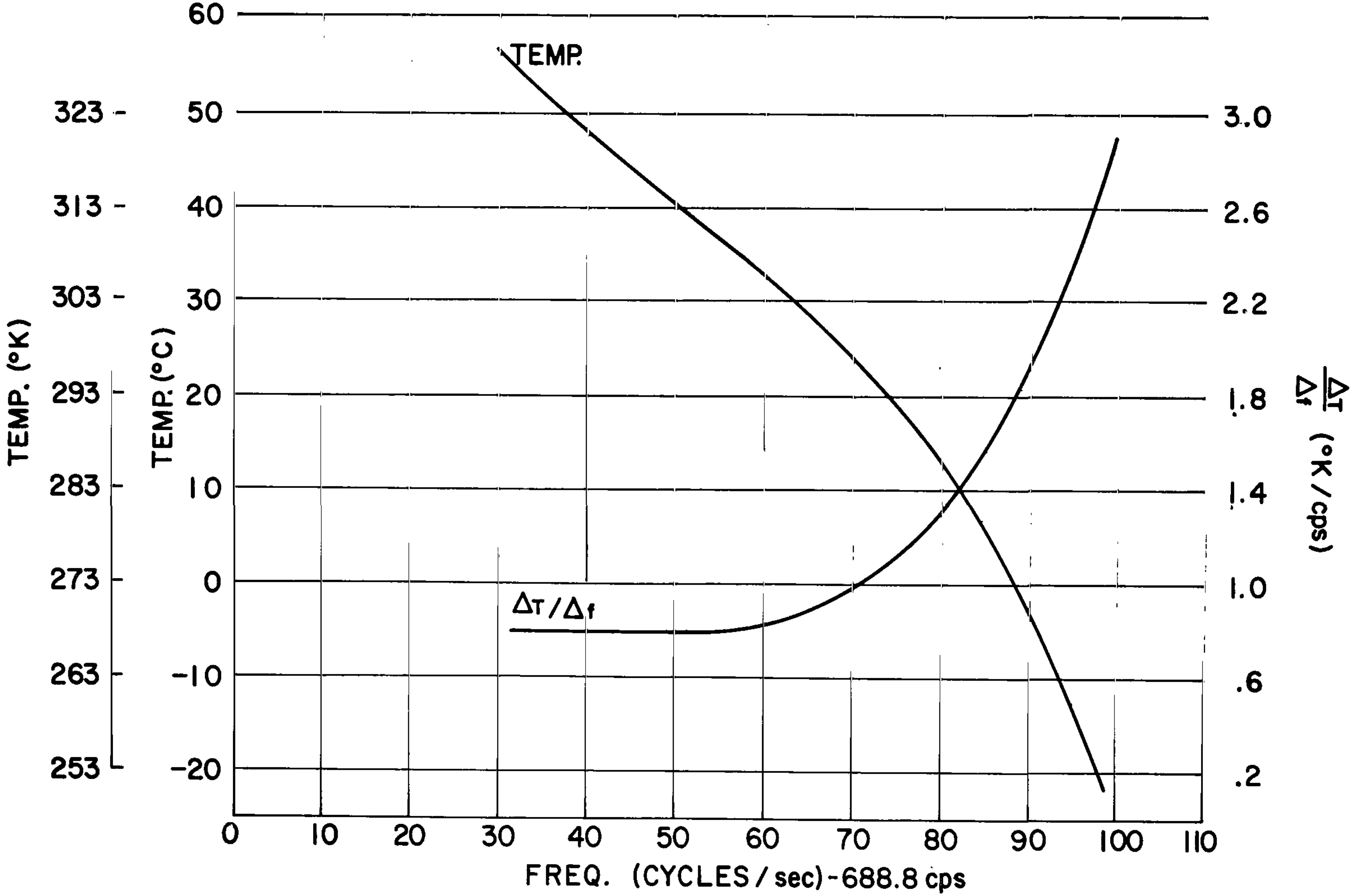


FIGURE 5. TEMPERATURE CONVERSION CHART

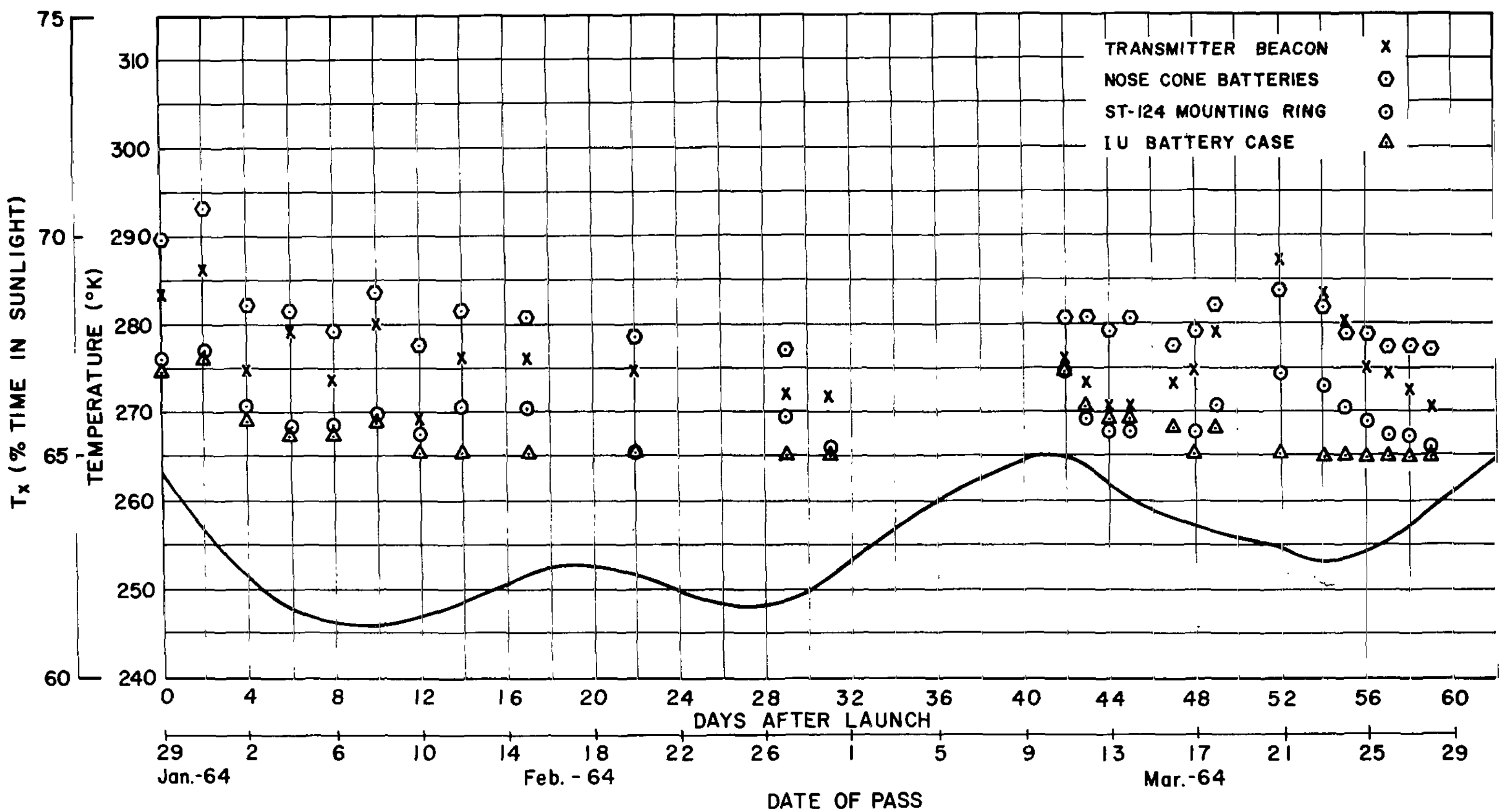


FIGURE 6. GREEN MOUNTAIN DATA

using the Gerber Variable Scale and the conversion chart in Figure 5. The data in Figure 6 show a change of approximately 10°K between the peaks of the temperature cycling from the second to the tenth day after launch. Due to the relatively high heat capacity of the payload, the temperature cycling during one orbit (94.86 minutes) varies only a couple of degrees Kelvin as the payload passes through the umbra and penumbra of the Earth's shadow. The percent time in sunlight (T_x) for the SA-5 payload is given in Figure 6, and was computed on the IBM 7090 computer by the MSFC Computation Laboratory. As the payload travels around the Earth in its elliptical (.0356 eccentric) orbit, it passes through the Earth's shadow during each revolution. The percentage of time that a satellite spends in the Earth's shadow during each revolution can vary from 0 to 40 percent. During the SA-5 telemetered orbital lifetime, the maximum time spent in the Earth's shadow during one orbit was 38.83 percent (February 10, 1964) and the minimum was 34.1 percent (April 3, 1964), the last day of telemetry from the SA-5 beacon transmitter. This shadow passage is important since it directly affects the temperature of the SA-5 payload. This time spent in the shadow during each revolution is continually changing due to the motion of the Earth about the Sun and the various orbital perturbations, primarily those caused by the oblateness of the Earth [1].

The points in the orbit at which the payload enters and leaves the Earth's shadow can be calculated using the shadow-sunlight computer program with the following input data:

T_o	Universal time during initial perigee passage (hr)
D_o	Days after vernal equinox
e	Eccentricity
LON	Longitude of initial perigee (° west)
LAT	Latitude of initial perigee
i	Inclination of orbit
R_p	Radius of perigee (km)
S	Sign of north velocity component at perigee (+ or -)
T	Other constants (such as the Kepler constants) which will be the same in all cases

Figure 7 is a graph showing results of the calculations computed on the IBM 7090 at MSFC to obtain the position of the Earth's shadow with respect to the orbit of the SA-5 payload (graphed with days after launch versus minutes from ascending node). The time is referenced to the ascending node because, generally, the times of nodal crossings are readily available.

The type of graph in Figure 7 is useful in planning the tracking of a satellite. A tracking station in a fixed geographic position can track only a certain part of each orbit. The two horizontal lines drawn parallel to each other across Figure 7 show the section of the orbit which can usually be seen by the Green Mountain Test Facility (Latitude: 34° 36' 47.39" N, Longitude: 86° 30' 57" W) at some time during each day since the payload must be in the direct line of sight to the receiving station. The payload enters the line of sight of the receiver at approximately 15 minutes from the ascending node, and exits the line of sight at about 30 minutes from the ascending node. Passes were taken only during normal working hours; therefore, no data were obtained on days when the payload was in the Earth's shadow during passage over Green Mountain because the receiving station was not operated in darkness.

The minimum and maximum temperatures obtained from the Green Mountain data are shown in Table I.

The NASA Space Tracking and Data Acquisition Network furnished six-channel sanborn strip charts, magnetic telemetry tapes, and reduced and teletyped quick-look temperature data from all minitrack stations. However, the first two and one-half weeks of the quick-look data was erroneous. Figures 8, 9, 10, and 11 present the minitrack quick-look data (the first group of points on each graph consisting of data points reduced by MSFC from the sanborn strip charts, in lieu of the erroneous quick-look data).

Post-launch calculated temperature maxima and minima were obtained using the actual orbital parameters. These results are given in Table I.

One important orbital parameter was unavailable – the orientation of the satellite with respect to the Sun.

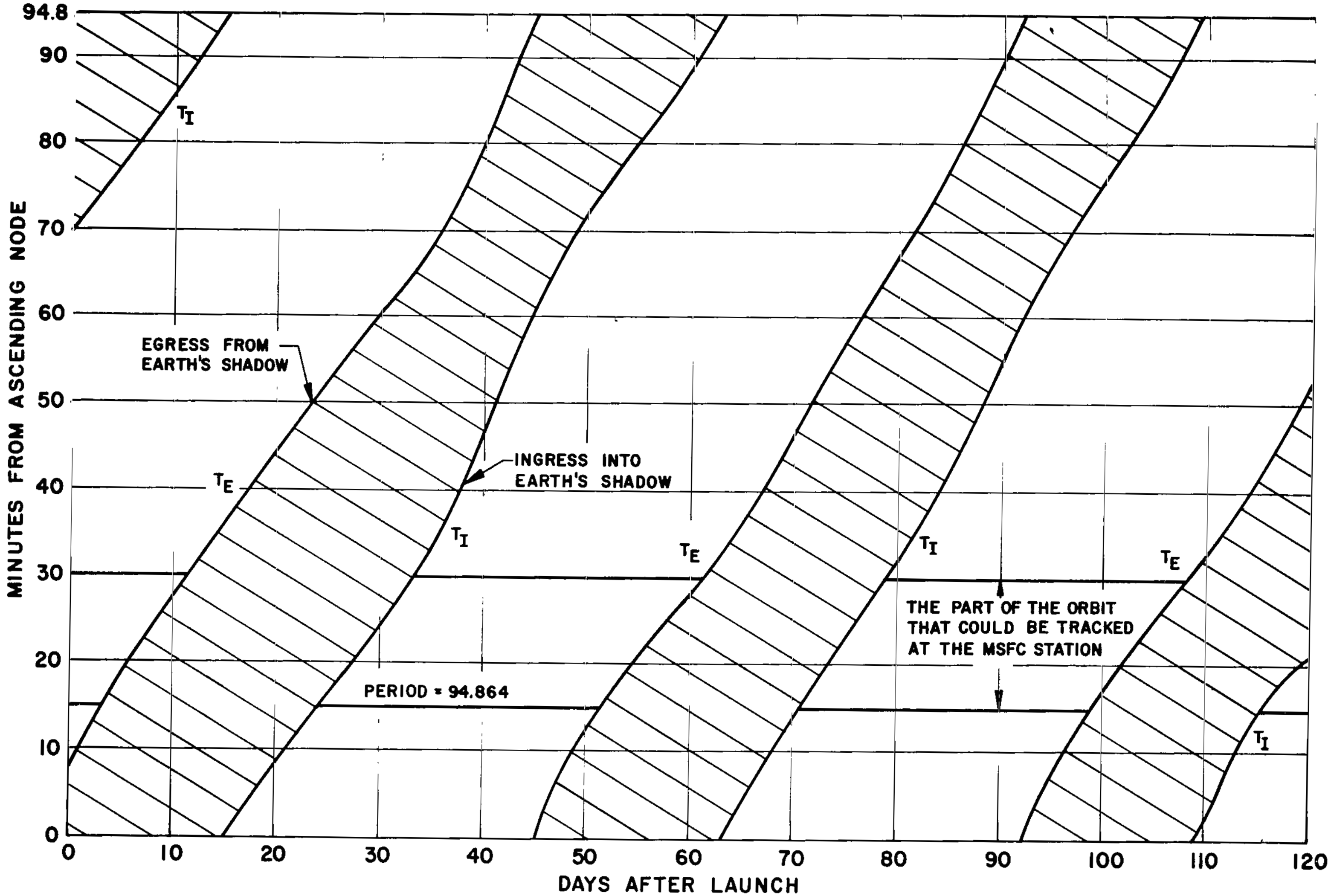


FIGURE 7. POSITION OF EARTH'S SHADOW IN ORBIT OF SA-5

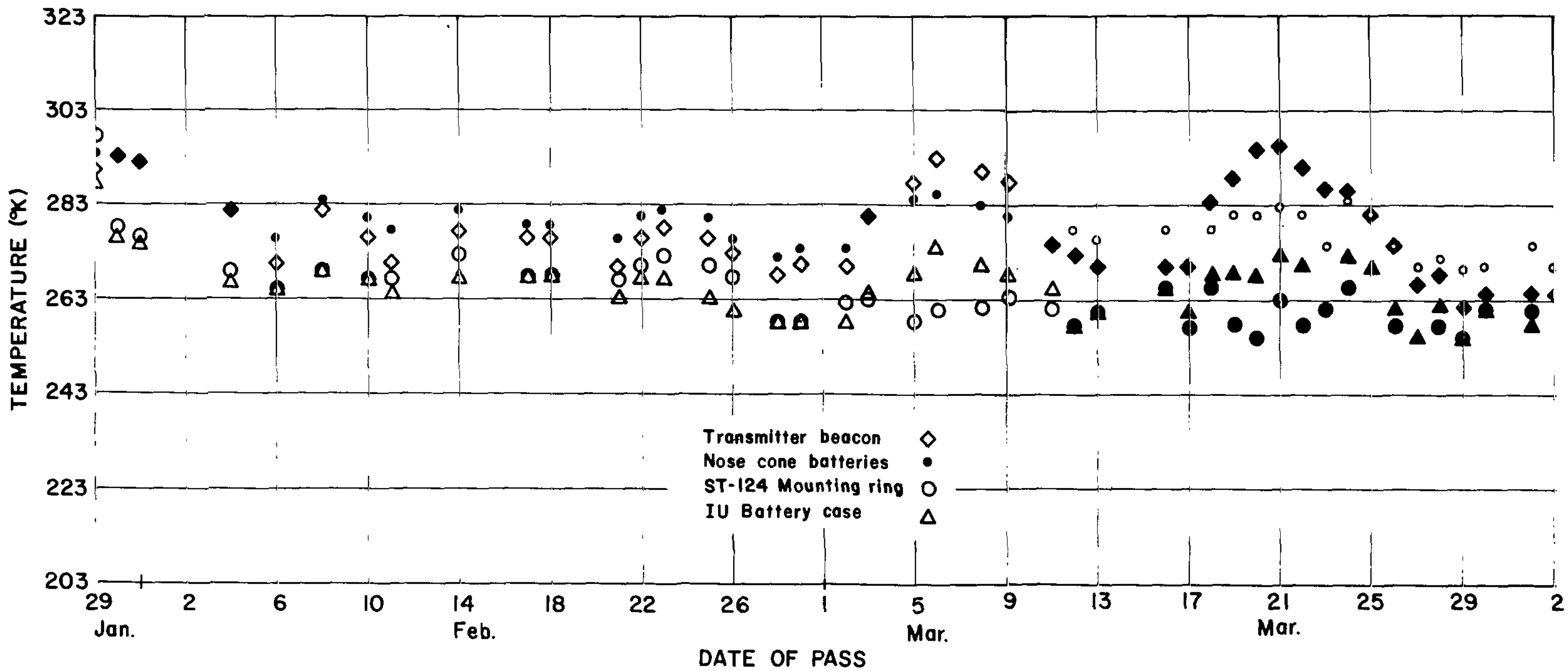


FIGURE 8. MINITRACK TEMPERATURES FROM SANTIAGO, CHILE (SA-5)

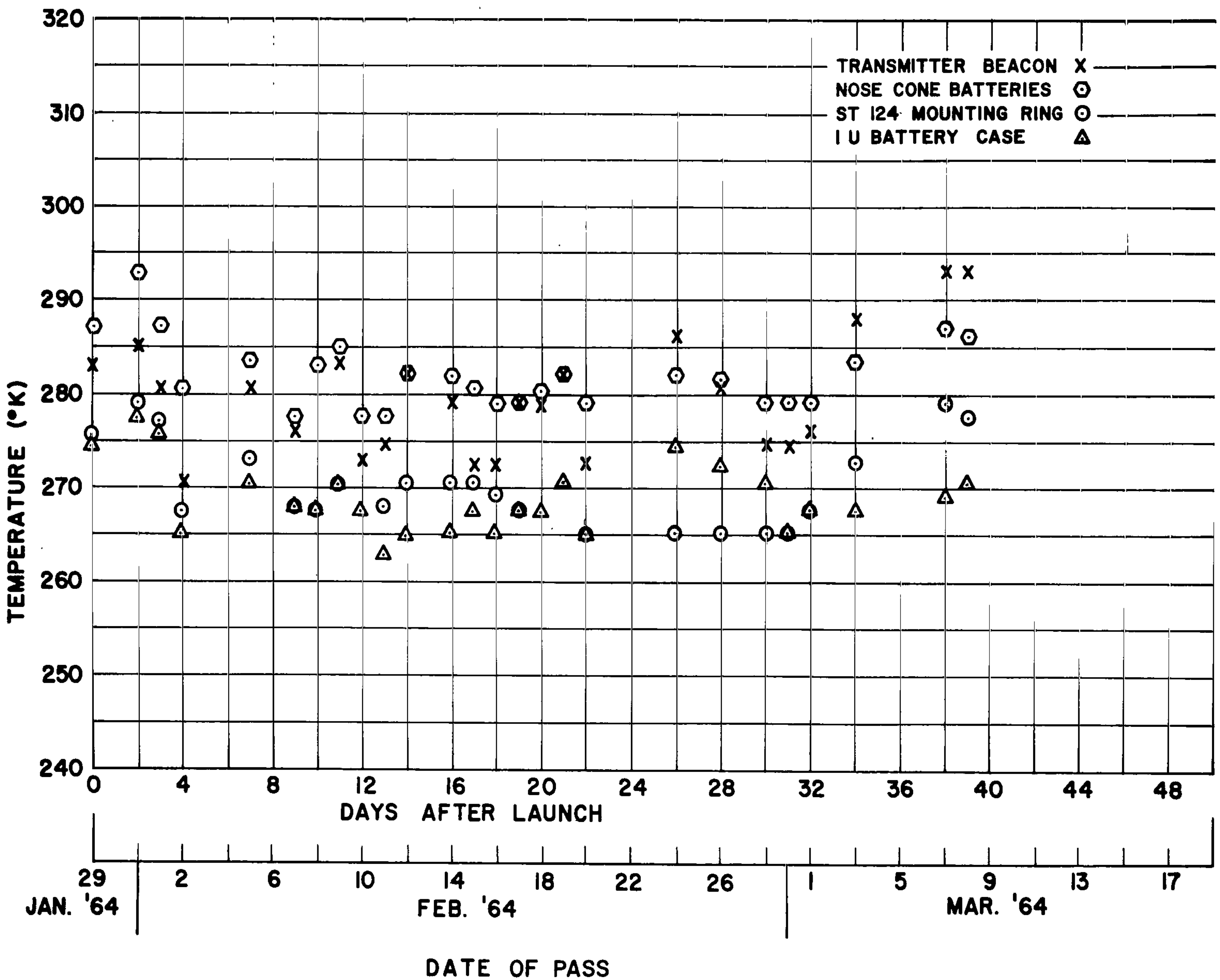


FIGURE 9. FORT MYERS DATA (GODDARD MINITRACK)

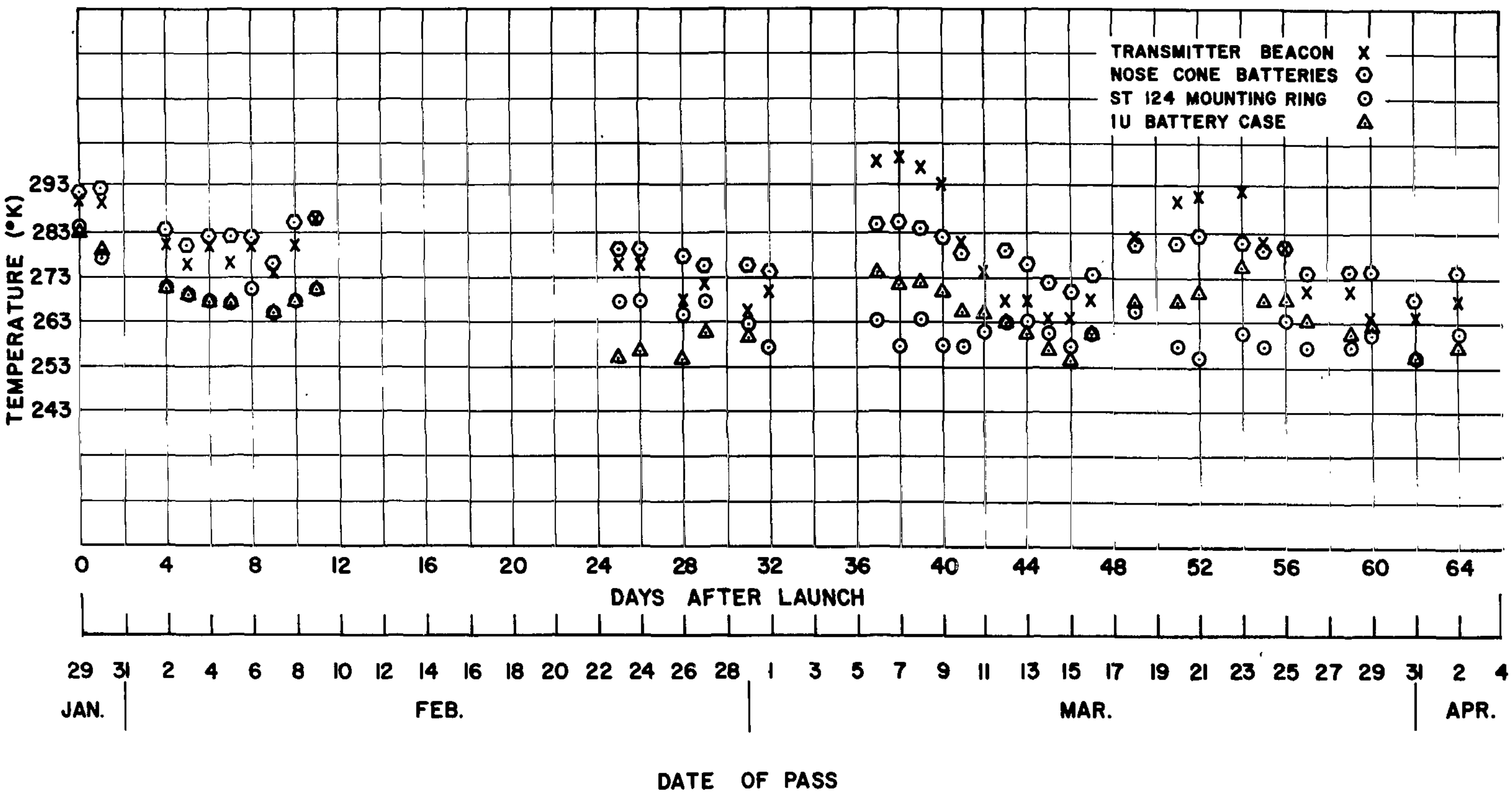


FIGURE 10. MINITRACK TEMPERATURES FROM LIMA, PERU (SA-5)

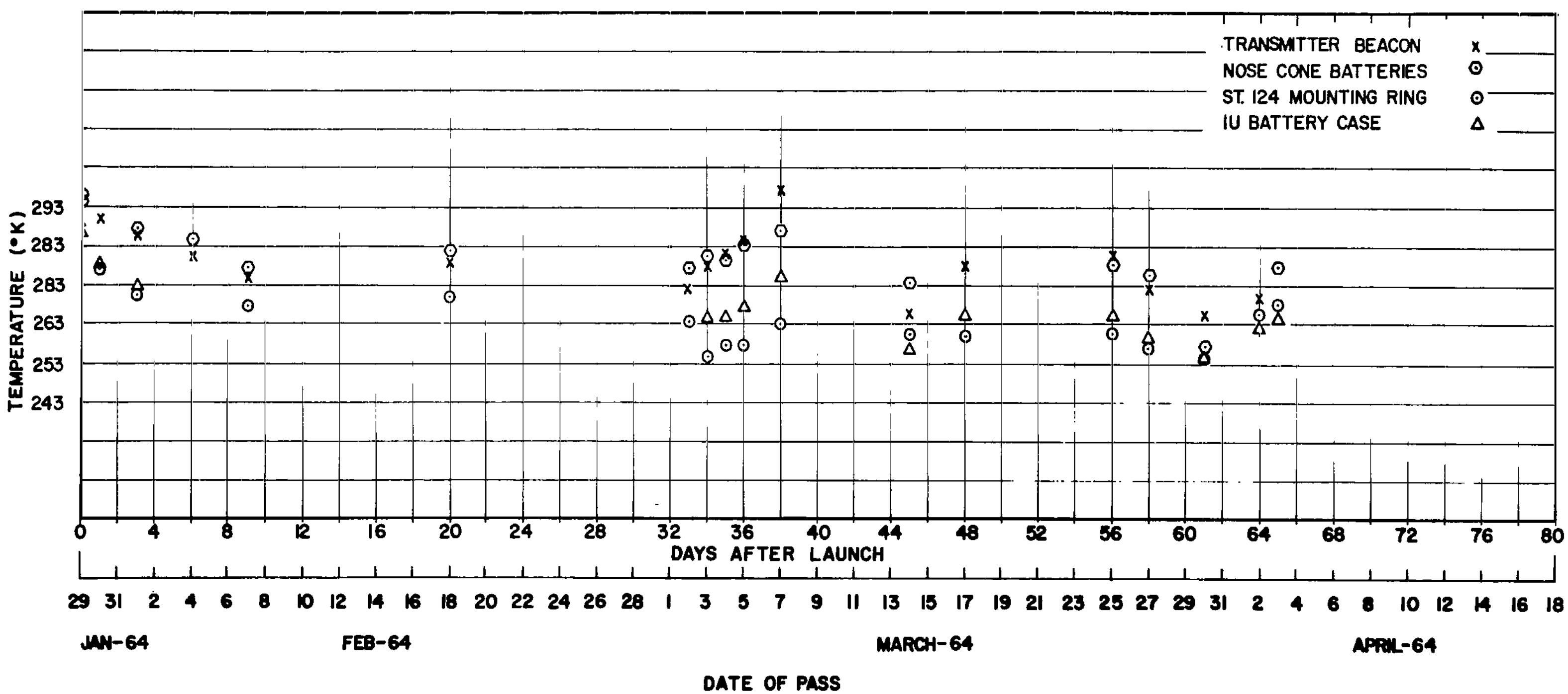


FIGURE 11. MINITRACK TEMPERATURES FROM JOHANNESBURG, SOUTH AFRICA (SA-5)

TABLE I

IN-FLIGHT MEASURED AND CALCULATED TEMPERATURES

Component	Specified Component Operating Temperatures		In-flight Measured Temperature Extremes		Calculated Temperatures	
	Maximum °K	Minimum °K	Maximum °K	Minimum °K	Maximum °K	Minimum °K
Nose cone batteries	348	253	293	277	295	273
Transmitter beacon	348	263	287	269	289	269
ST-124 mounting ring in IU	None	None	277	265	276	257
IU battery case	None	None	276	265	276	257

An analysis was performed in order to relate the payload orientation with respect to the Sun and the measured temperatures. The payload temperatures are a function of the solar energy absorbed, which is a function of the payload projected area with respect to the Sun. In this study the nose cone battery temperatures are utilized. For this measurement, the sensitivity of the telemetered data is higher because $\Delta T/\Delta f$ is consistently less (Fig. 4). Also the calculated temperatures of the nose cone batteries are more accurate since this calculated temperature is independent of the IU whose solar absorptance is unknown. The various orbital parameters for a tumbling satellite were obtained and the temperatures were computed for three projected areas including the maximum and minimum projected areas for the flat tumble mode. These were plotted with the measured temperature, superimposed, and are shown in Figure 12. The parameter, %A, is defined as the percent projected area of the maximum projected area.

Another interesting fact concerns the degradation of the IU exterior coating (which was not a thermal control coating). The coating is a white paint known as "camouflage white" (TiO_2 in alkyd).

A solar absorptance-infrared emittance ratio of approximately .67 is required to explain the IU temperatures which are approximately double the initial ratio.

REFERENCE

1. Krause, H. G. L., "Secular and Periodic Perturbation of the Orbit of an Artificial Earth Satellite," translated from German by Seymour Nelson – paper presented at the VII International Astronautical Congress in Rome, Italy, September 17-22, 1956.

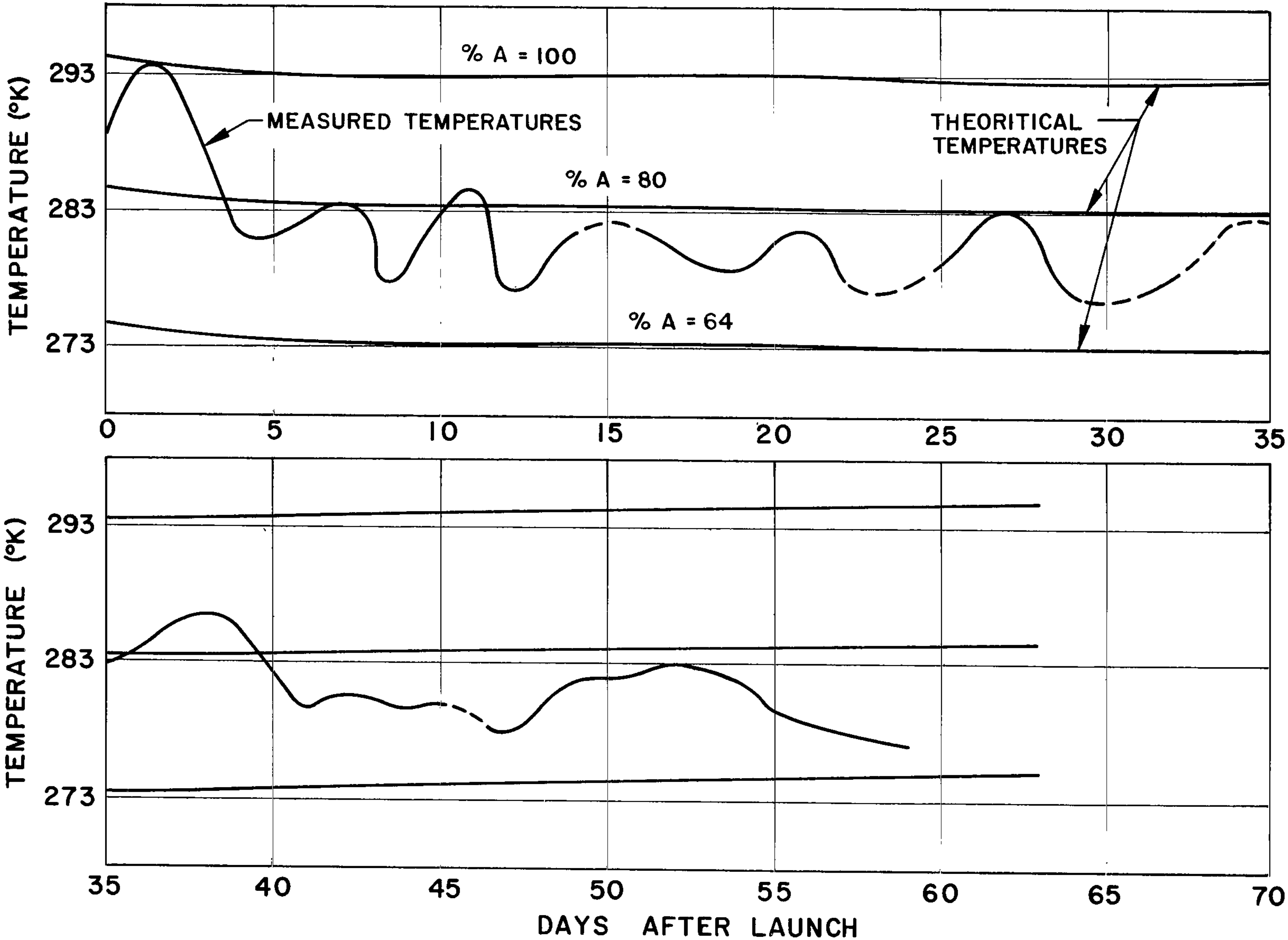


FIGURE 12. NOSE CONE TEMPERATURE VERSUS DAYS AFTER INJECTION INTO ORBIT. THREE THEORETICAL CURVES ARE SHOWN FOR VARIOUS PROJECTED AREAS OF THE PAYLOAD TO THE SUN

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